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DEVELOPMENT OF A HYDROGEN-BURNING ANNULAR COMBUSTOR FOR USE IN A MINIATURE GAS TURBINE ENGINE

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SUMMARY

This report covers the completed work of NASA Statement of Work L53-9008 dated March 25, 1968 on the development of a Hydrogen-Burning Annular Combustor for use in a Miniature Gas Turbine Engine for wind tunnel test programs. The burner was designed for a 4:1 pressure ratio engine, 3.6 inch inlet diameter using 1.87 pounds of air per second and an average temperature rise of 1500°F. The annular burner successfully demonstrated an average temperature rise of 1500°F. minimum and showed little deterioration after a 5 hour 24 minute endurance test at the above conditions.

INTRODUCTION

Propulsion simulation in wind tunnel testing has proven to be extremely useful in obtaining aircraft aerodynamic characteristics. The present simulation techniques using externally supplied high pressure gas to power either turbine driven compressors simulating fan jet engines or ejector powered jet-engine simulators fail to simulate both inlet and exit conditions. One method of simulating an aircraft gas turbine engine in wind tunnel testing (especially temperature effects) is to use a scaled down version of the gas turbine engine. However, development of small gas turbine engines in the past has been limited by the inability to scale down the length of the combustion section to match the burning characteristics of the available fuels.

The fuel must be completely burned within the combustion liner, therefore the fuel-air mixing and burning rate determines the length and volume of the combustion liners. A possible solution to this problem was to use a fuel having a fast burning and mixing rate. Some preliminary work was done on a small two-dimensional burner using gaseous hydrogen fuel which showed that burning could be accomplished in a short combustor. As a result of these preliminary results the present investigation was undertaken to develop a complete hydrogen-burning annular combustor suitable for use in a miniature gas turbine engine.

The combustor program consisted of developing a fuel nozzle and combustion liner combination which optimizes combustion efficiency, and minimizes pressure drop and exit temperature variation while achieving a minimum ΔT of 1500°F. across the burner.

LIST OF SYMBOLS

SYMBOL	DEFINITIONS
C_p	specific heat at constant pressure - BTU/lb°F
F	thrust - pounds
g	gravitational acceleration 32.2 ft./sec.^{2}
h	enthalpy per unit mass - BTU/lb.
ΔH	lower heating value of hydrogen 51574 BTU/lb.
j	778.16 ft. - lb./BTU
m	mass - pounds
P	pressure - psi
ΔP	differential pressure - psi
P_{r_0}, P_{r_1}, etc.	relative pressure
°R	degrees Rankine
T	temperature °F
ΔT	differential temperature °F
ΔTVR	$\frac{\text{max. local comb. outlet temp.-aver. comb. inlet temp.}}{\text{aver. comb. outlet temp.-aver. comb. inlet temp.}}$
V	velocity - ft./sec.
W_a	flow - pounds - air/sec.
W_{ea}	flow - pounds excess air/sec.
W_f	flow - pounds fuel/sec.
η	efficiency

SUBSCRIPTS	DEFINITIONS
0	at engine inlet
3	at combustor inlet
4	at turbine inlet
5	at turbine outlet
8	at jet nozzle throat
'	after efficiency considerations

CHEMICAL SYMBOLS	DEFINITIONS
H_2	Hydrogen
H_2O	Water or steam
O_2	Oxygen
N_2	Nitrogen

TEST INSTALLATION

The air supply for this program was provided by an Airesearch TMC 105 gas turbine Start Cart. The air was piped to the test section by a 3.5-inch pipe and was cooled enroute by two in-line single-pass water heat exchangers to a nominal 320°F. at the burner inlet. Air flow was measured in the air line by use of an orifice plate. The hydrogen was stored in a truck trailer at 2500 psi. The burner section was mounted in a vertical position in a fenced in area outside of the test building. Remote control for operating the test were located inside the test building. Figure 1 shows the general layout of the test set up while Figure 2 is a photograph of the burner installed and in operation.

Instrumentation

Airflow rates were measured by a square-edged orifice installed according to ASME specifications. The ΔP across the orifice plate was read on a mercury "U" tube manometer. Air temperature for the flow calculations and for the combustion chamber inlet were measured with Iron-constantan thermocouples and read on a Honeywell multi-channel dial type instrument. Combustor exit air total temperature was measured with Chromel-Alumel ceramic insulated thermocouples on a recording chart type "Honeywell-Brown Electronik" instrument.

Hydrogen temperature for the flow calculations was measured with a Copper-Constantan thermocouple on a digital indicator. Inlet air total pressure was measured with a three-tube equal-area rake and read on standard dial type gages. Combustor exit total pressure was measured around the annular nozzle with a moveable total pressure probe.

CALCULATIONS

The characteristics of the combustor are calculated as follows and the results are presented in Table 1.

Combustion efficiency was defined as the ratio of actual temperature rise to theoretical temperature rise

$$\eta_c = \frac{\Delta T\ (\text{actual})}{\Delta T\ (\text{theoretical})}$$

$$\text{Where } \Delta T\ (\text{theoretical}) = \frac{\Delta H}{(m\ C_p)_{H_2O} + (m\ C_p)_{N_2} + (m\ C_p)_{AIR}}$$

and the values for (m) are obtained from the chemical equation

$$H_2 + 1/2\ O_2 + 1/2\ (3.76)\ N_2 + W_{ea} \rightarrow H_2O + 1/2(3.76)N_2 + W_{ea}$$

$$(1) + (8) + (26.34) + (92.41)^* \rightarrow (9) + (26.34) + (92.41)$$

*based on air fuel ratio = 126.74

Excess air is calculated as follows

$$W_{ea} = \frac{W_a\ (\text{actual})}{W_f\ (\text{actual})} - \frac{W_a\ (\text{theo})}{W_f\ (\text{theo})}$$

$$= \frac{W_a\ (\text{actual})}{W_f\ (\text{actual})} - 34.33\ \frac{\text{lbs.-air}}{\text{lbs.-fuel}} = 126.74-34.33$$

$$= 92.41\ \text{lbs. excess air/lb. fuel}$$

Combustor pressure loss $\Delta P/P$ was defined by the following expression:

$$\frac{\Delta P}{P} = \frac{\text{aver. inlet total pres. - aver. exhaust total pres.}}{\text{aver. inlet total pressure}}$$

The reference velocity used in this report, which is a common term in the gas turbine industry, is based on maximum cross-sectional area of combustor housing flow path and the static pressure and temperature at the inlet to the combustor.

COMBUSTORS

Three major combustion liner configurations and four major fuel nozzle configurations were tested plus several variations of each for a total of 65 tests. The liner is annular, 1.80 inches long and is constructed of .035 inch thick "Hastelloy X." Some of the configurations tested in the development program are shown on Figures 3, thru 8. The first liners were built with air holes and cooling louvers as indicated by configuration runs 16 and 32 shown on Fig. 3. The louvers were later abandoned for additional holes. The first fuel nozzles were tubes and were separable from the liner, see photograph on Figure 9. This design was changed after much testing for an integral nozzle and combustor liner (see Figure 4 to 8.) The final configuration is that indicated for runs 56 to 65 on Figure 8. A sketch showing the final burner assembly is shown in Figure 10. Most of the changes in design were in an effort to improve the circumferential temperature distribution and to prevent burning of the liner. An automotive type 1/4 inch spark plug was used for ignition.

TEST CONDITIONS AND PROCEDURES

The object of the tests was to obtain a ΔT across the burner of 1500°F. with a flow of 1.87 lbs./sec. and a 4:1 pressure ratio with inlet air temperature to the burner of 320°F. The runs which did not reach these conditions were cut short in most cases due to hot spots above 2500°F. which were destructive to the liner. Figure 11 shows the damage which results from hot spots in the liner.

The procedure for operation of the burner test was as follows:

1. Inlet air pressure adjusted to about 1 or 2 psig

2. Ignitor turned on

3. Vernier fuel flow valve opened

4. When temperature jumped (600°F. to 800°F.) air flow valve was gradually opened to full open position, the spark plug was shut off, at the same time the main fuel valve was opened at a rate sufficient to prevent temperatures over 2200°F.

5. The fuel flow, combustor air inlet pressure and inlet air temperature were adjusted to desired conditions.

6. Data recorded

7. To shut down, the fuel valve and then the air valve were turned off.

RESULTS AND DISCUSSIONS

The many combustor configurations were evaluated according to the following criteria: combustor outlet temperature distribution, liner life, combustion efficiency and pressure loss. The combustor temperature distribution became the major development problem of the program. Wide variations in temperature around the annulus occurred on most of the early designs. It is important to obtain equal temperature distribution around the burner annulus for two reasons: first, variations in temperature cause structural problems due to varying thermal growth, second, the average 1500°F ΔT required for a gas turbine engine to run means that temperature peaks much in excess of the average can cause burning of the liner and/or the turbine nozzle and turbine wheel.

The liner life initially was a problem because of hot spots. A typical burned liner is shown in Figure 11. The arrangement of air holes in the liner and fuel holes in the nozzle determine the fuel distribution and the existence of hot spots or streaks.

Combustion efficiency was not a problem in this development program. It is of course a measurement of how much burning is taking place in the combustor itself.

Pressure loss in the burner section is a function of the liner design. In this development program pressure drop through the liner was not given much emphasis until after the temperature distribution problem was solved. Further development work should easily reduce the pressure drop.

Table 1 shows a complete run down of the important data and parameters of all the significant configurations tested. Results which show combustion efficiency over 100% have been attributed to improper sampling.

After several shake down tests of the facility and its instrumentation a full data run, number 16, (Fig.3) was completed. The average ΔT across the burner was low as can be seen in Figure 12 due to choking of the fuel in the fuel nozzle inlet tube. Figure 9 is a photograph of a typical tube fuel nozzle. The liner in this configuration was louvered similar to aircraft gas turbine combustion liners (see Fig.11.) Figure 11 is a photograph of a burned liner similar to the liner in run 16.

The inlet to the fuel nozzle was enlarged and efforts were made to have all nozzle holes precisely the same diameter. After many tests on several nozzle hole diameters and arrangements, it was found that the hydrogen would not distribute evenly regardless of the ΔP across the nozzle. It had been presumed that if choked flow at the nozzles could be reached each nozzle hole would deliver the same flow providing the holes were of identical diameter. The tests which were run proved that this is not necessarily true. A typical test point indicating the results of these nozzle tests is run 32 (Fig.3). Figure 12 shows the the circumferential temperature distribution obtained at the simulated turbine nozzle exit. All of the tests through run 36 used a stainless steel tube as the plenum for the fuel nozzle.

Having used as large a tube as possible without improving the fuel distribution the configuration shown for run 37 (Fig.4) was constructed which allowed considerable more plenum volume. The deflector burned off during run 37 and was retested as run 38 without a deflector. Temperature distribution charts for these runs are shown in Figures 13 and 14. The problem of distribution was not solved with the larger plenum. In run 41 and 42 (Figure 5) an attempt to solve the distribution problem with a porous material in place of drilled holes was made. The nonhomogeneity of the porous material nulified any improvements it might have made. The temperature distribution for run 42 is shown in Figure 15. Further attempts to make the porous material work were tried in run 43 with a new liner. The liner had drilled holes in place of louvers (see Fig. 5, Run 43.) It was thought that the louvers may be allowing unequal quantities of air to pass through the various circumferential positions. An attempt was made with the porous material in run 44 (Fig. 6) to swirl the air immediately aft of the porous nozzle in an effort to even out the high temperature streaks associated with the uneven hydrogen distribution. See Figure 16 for a photograph of this liner. The hot streaks persisted with this design. It was concluded that since gas entering a limited volume plenum from a single source will not distribute evenly among the many nozzle holes a multi entrance system of the gas to the plenum would have to be devised. It was reasoned that a double plenum chamber as shown on the combustor assembly in Figure 10 with an equally spaced set of holes at the interface would greatly reduce the effects of the uneven distribution. This reasoning was proven correct in run 50 (Fig. 6.) The temperature variation (ΔTVR) showed considerable improvement with a value of 1.17 (See Table I.) A value of 1.00 indicates no temperature variation around the exit.

With the temperature distribution problem solved efforts were made to reduce the pressure loss through the liner. Larger holes were drilled in the liner resulting in reduced pressure loss (ΔP)/P on run 54 (Fig. 7 & Table I.) Run 55 (Fig. 7) had an additional row of holes in the outer liner which resulted in cold streaks and a poorer ΔTVR as indicated by the circumferential distribution of temperature of Figure 17. Runs 56 and 57 (Fig. 7) had the size of the selected forward rows of liner holes increased which resulted in the final configuration. Runs 58 and 59 were the same as the previous run except it had a new outer liner. The old liner which had been cut and welded many times as well as subjected to several runs with temperatures in excess of 2500°F. was warped. The temperature distribution for run 58 (finalized configuration) is shown in Figure 18. Figure 19 is a photograph of the final combustor before the welding of the outer liner to it.

A five hour endurance run was completed (runs 60-62) without repairs at rated temperature, flow and pressure (Figures 8 & 20.) The endurance run was stopped twice for visual checks of the liner and once for a switch over of hydrogen storage trailers.

The entire run was completed in less than seven hours. The five hours of running at rated conditions appear to be only a fraction of the liner's total life. See Figures 21 and 22 for before and after photographs of the liner. Figure 23 shows the condition of the simulated turbine nozzle following all 65 tests done on this program.

The final combustion liner design is capable of being used in the gas turbine engine as originally proposed. Using the combustor design of this development program and compressor and turbine design each of 75% efficiency, a thrust of 100 pounds can be expected from an engine with a 3.6-inch inlet diameter flowing air at 1.87 pounds per second. A proposed gas turbine engine designed for the above performance is shown on Figure 24. The predicted engine performance is shown in the Appendix.

APPENDIX

Predicted Engine Performance

Inlet temperature	T_o	$= 520°R$
from Keenan & Kaye*	P_{ro}	$= 1.2147$
" " "	h_o	$= 124.27$ BTU/lb.
for 4:1 pressure ratio	P_{r_3}	$= 4\ (1.2147) = 4.8588$
from Keenan & Kaye	T_3	$= 771.5°R$
" " "	h_3	$= 184.88$
compressor work	Δh_c	$= 184.88-124.27=60.61$ BTU/lb.
assume 75% efficiency	$\Delta h'$	$= \frac{60.61}{.75} = 80.813$ BTU/lb.
enthalpy level at comb. inlet	h_3'	$= 124.27 + 80.81=205.08$ BTU/lb.
combustor inlet temp from Keenan & Kaye	T_3	$= 854.4°R$
combustor temperature rise	ΔT	$= 1500°F$
combustor exit temperature	T_4	$= 2354.4°R$
from Keenan & Kaye	h_4	$= 604.24$ BTU/lb.
turbine work (75% efficiency)	Δh_T	$= \frac{80.81}{.75} = 107.75$ BTU/lb.
from Keenan & Kaye	P_{r_4}	$= 339.5$
enthalpy level at turbine exit	h_5'	$= 604.24-107.75=496.49$ BTU/lb.
from Keenan & Kaye	P_{r_5}	$= 163.46$

from Keenan & Kaye $T_5 = 1970.3°R$

turbine exit pressure (assume P_4 = 3.8 atmos.) $P_{T5} = P_4 \dfrac{P_{r5}}{P_{r4}} = 1.829$ atmos.

exhaust nozzle exit temperature $T_8 = \left[\dfrac{P_{T5}}{P_o}\right]^{-\left[\frac{\gamma-1}{\gamma}\right]} \left[T_{T5}\right]$

$= 1.829^{-0.245283} \quad 1970.3$

$= 1699°R$

velocity at exhaust nozzle exit $V_8 = \sqrt{2\ g\ J\ c_p\ T_{T5} \left[1 - \dfrac{T_8}{T_{T5}}\right]}$

$= 1850$ ft./sec.

$F = \dfrac{V_8}{g}$

$= 57.45$ lbs thrust/lb air/sec.

$F = 57.45\ (1.87) = 107.4$ lb thrust

* GAS TABLES - Thermodynamic Properties of Air Products of Combustion and Component Gases Compressible Flow Functions by Joseph H. Keenan and Joseph Kaye - John Wile & Sons, Inc.

TABLE I - COMBUSTOR PERFORMANCE CHARACTERISTICS

Run-Point	$\frac{W_f}{W_a}$	$\frac{W_a}{W_f}$	W_f lb/sec	Theo ΔT	Act. ΔT Aver.	η comb. %	W_a lb/sec	Ref. Vel. ft/sec	ΔTVR	$\frac{\Delta P}{P}$ %	T Inlet °F	T exit aver. °F
5-3							2.200	187.5	2.350	9.70	335	
16-1	.00432	231.0	.00731	848	541	63.8	1.690	167.8	2.800	7.48	330	871
32-1	.00820	121.9	.01436	1554	1180	75.9	1.751	146.5	1.330	2.83	331	1511
37-1	.00895	111.7	.01530	1685	1201	71.2	1.710	143.6	1.443	3.36	307	1508
38-1	.01350	74.1	.01752	2441	1229	50.0	1.300	106.0	1.430	3.36	303	1532
41-1	.00777	128.7	.01350	1478	1136	76.9	1.740	145.1	1.469	2.53	316	1452
42-1	.00793	126.0	.01360	1507	1161	77.0	1.714	138.5	1.463	2.53	292	1453
43-1	.00767	130.4	.01300	1460			1.700			2.52		
44-1	.00534	187.3	.01000	1039	922	88.7	1.880		1.970		330	1252
50-1	.00832	120.2	.01380	1575	1600	101.6	1.660	139.6	1.170	28.54	326	1926
54-1	.00829	120.6	.01386	1570	1582	100.8	1.670	136.2	1.215	13.49	322	1904
55-1	.00786	127.2	.01320	1495	1482	99.1	1.680	143.7	1.303	16.54	343	1825
56-1	.00830	120.5	.01390	1572	1569	99.8	1.680	139.6	1.340	14.12	328	1897
57-1	.00807	123.8	.01360	1533	1577	102.8	1.684	137.6	1.341	14.3	315	1892
58-2	.00837	119.5	.01510	1584	1565	98.8	1.804	140.0	1.139	12.61	338	1903
59-1	.00817	122.4	.01504	1550	1578	101.8	1.840	141.4	1.145	12.23	318	1896
60-1	.00771	129.7	.01444	1468	1511	102.9	1.873	142.7	1.142		325	1836
60-2	.00800	125.0	.01480	1519	1554	102.3	1.850	140.6	1.121	12.58	323	1877
60-3	.00826	121.1	.01530	1564	1605	102.6	1.853	140.8	1.126	11.64	323	1928
60-4	.00810	123.5	.01530	1536	1568	102.1	1.889	141.4	1.124	12.03	323	1891
60-5	.00810	123.5	.01530	1536	1571	102.3	1.890	140.5	1.107	11.17	320	1891
60-6	.00810	123.5	.01540	1536	1602	104.3	1.890	141.8	1.108	11.84	325	1927
60-7	.00820	121.4	.01540	1561	1602	102.6	1.870	141.1	1.106	11.93	323	1925
60-8	.00848	117.9	.01535	1603	1597	99.6	1.810	138.9	1.110	11.81	327	1924
61-1	.00831	120.3	.01543	1573	1582	100.5	1.857	139.7	1.089	12.59	327	1909
61-2	.00833	120.0	.01563	1577	1605	101.8	1.876	140.4	1.118	12.29	325	1930
62-1	.00810	123.5	.01535	1537	1552	100.9	1.894	143.1	1.099	12.03	330	1882
65-1	.00901	110.9	.01560	1696	1281	75.5	1.698	125.1			309	1590

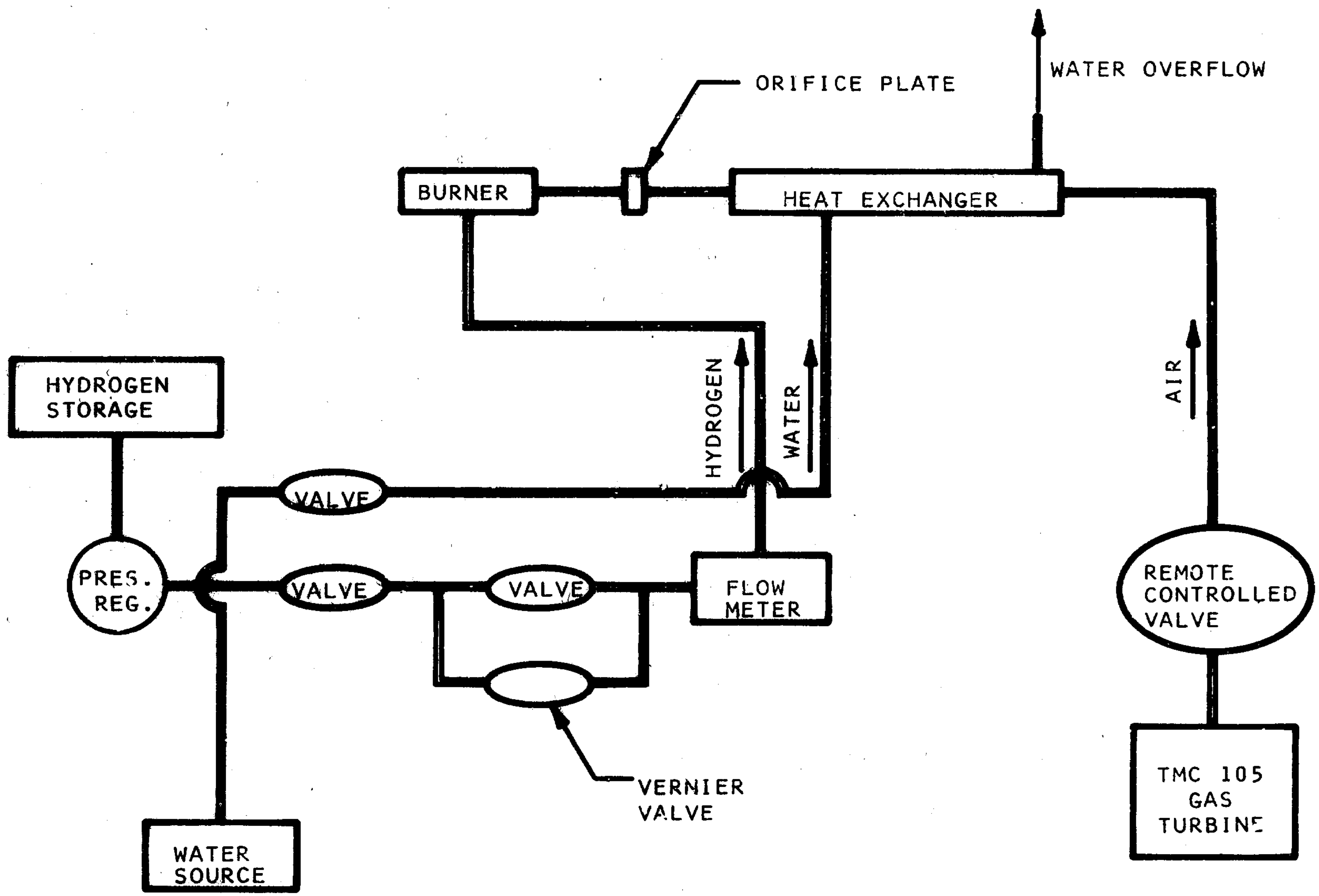


FIGURE 1 - TEST INSTALLATION SCHEMATIC

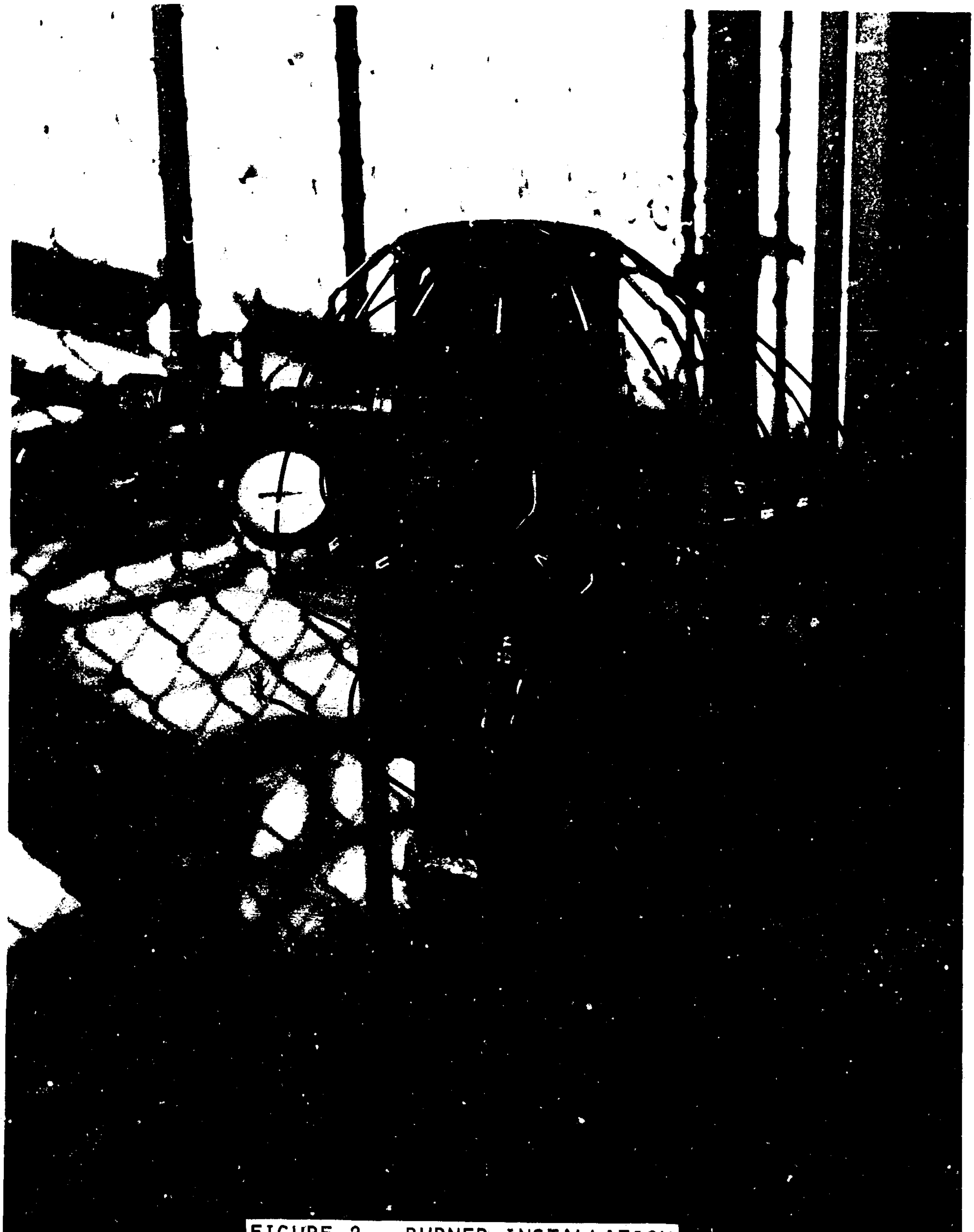
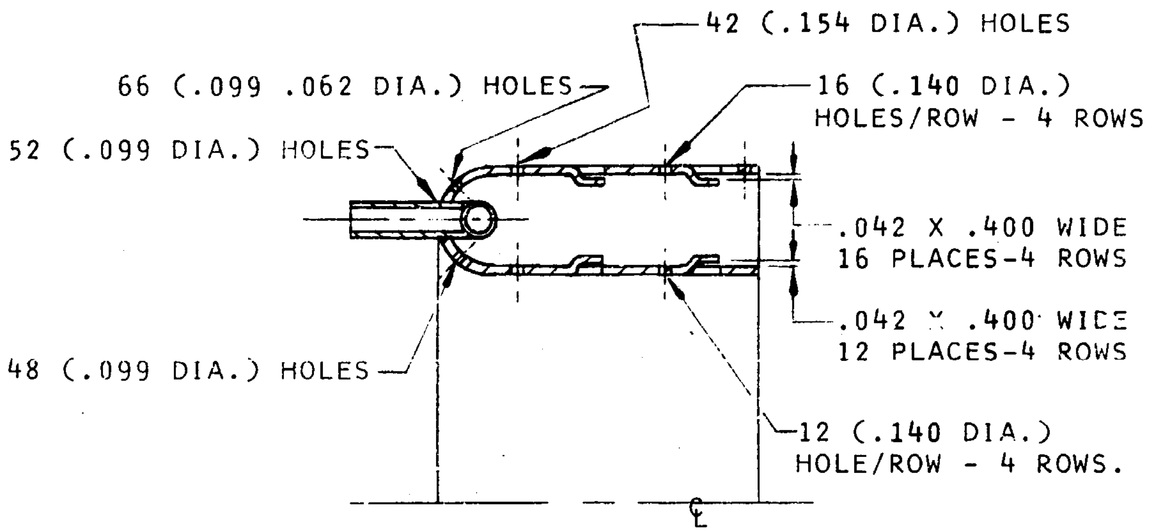
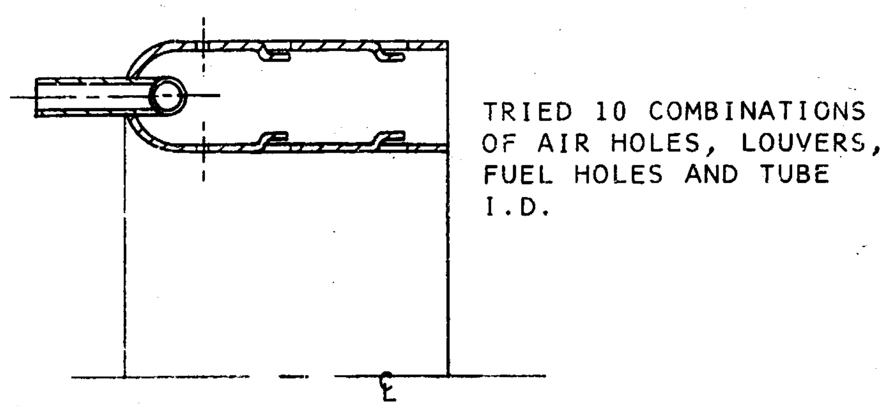


FIGURE 2 - BURNER INSTALLATION

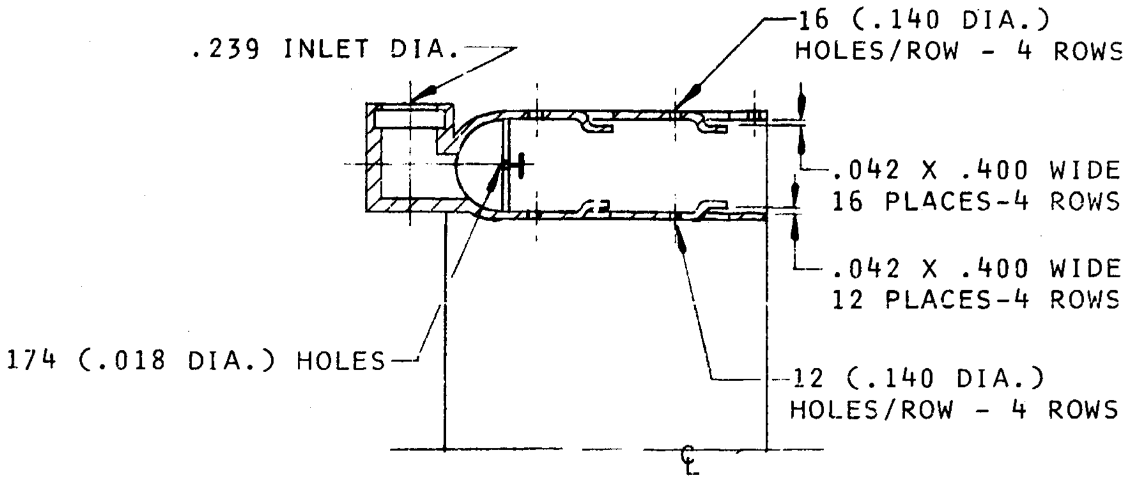


RUN 16

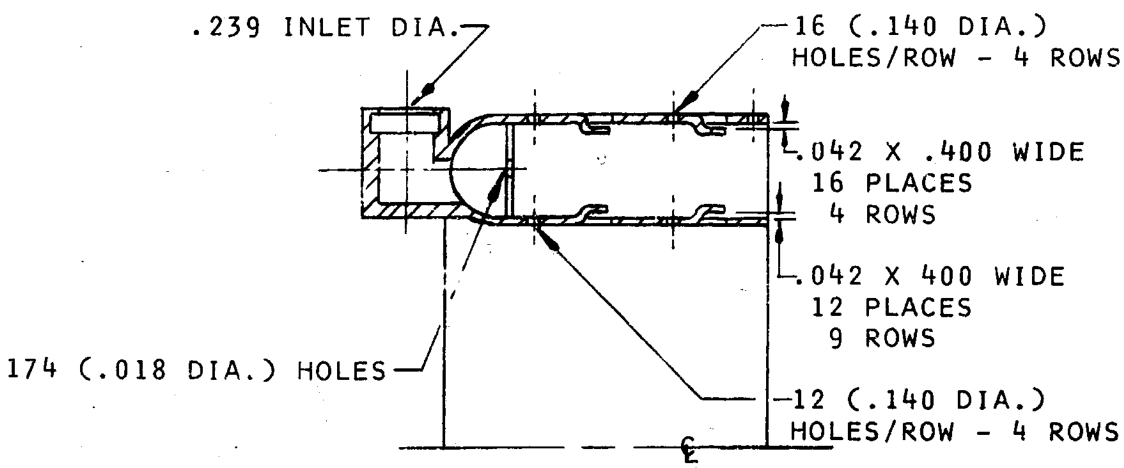


RUN 32

FIG. 3

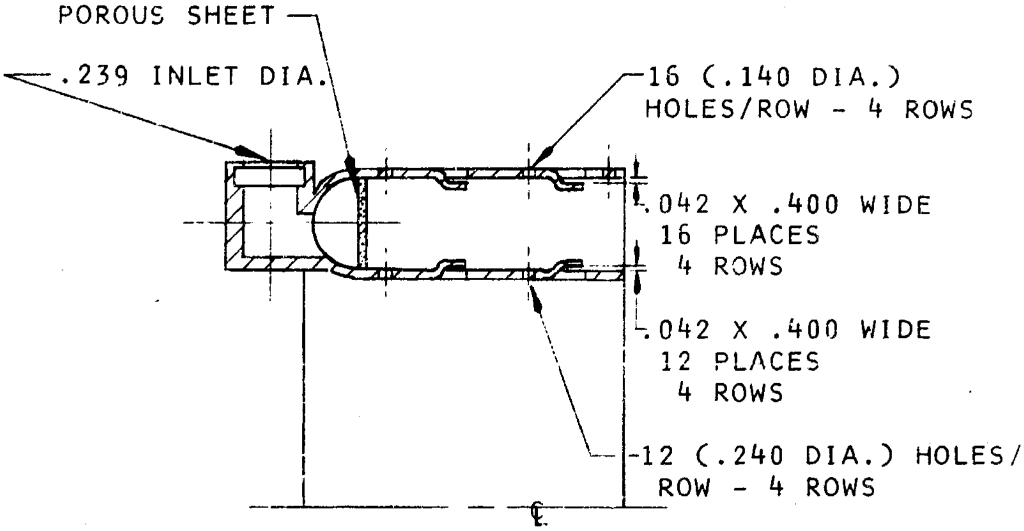


RUN 37

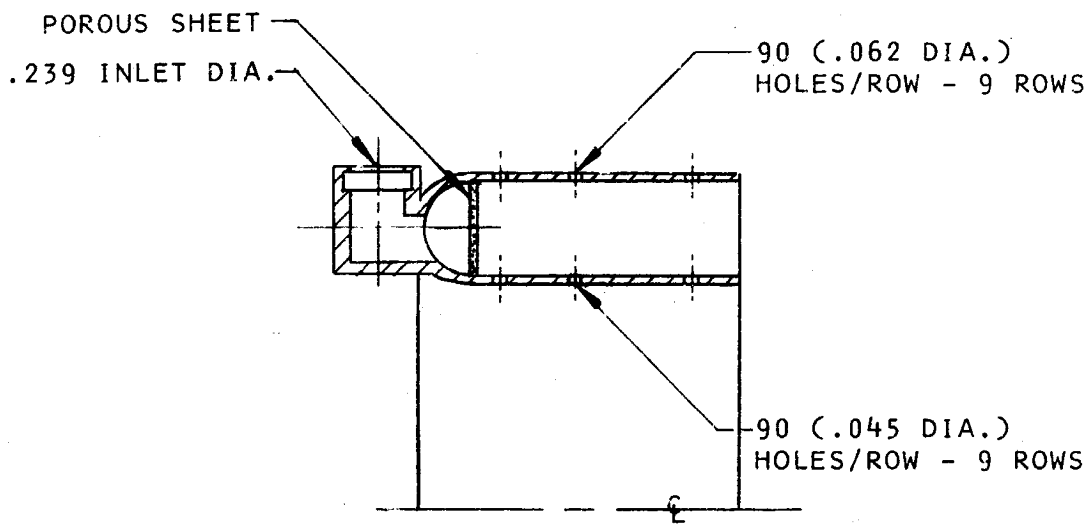


RUN 38

FIG. 4



RUN 41 - 42



RUN 43

FIG. 5

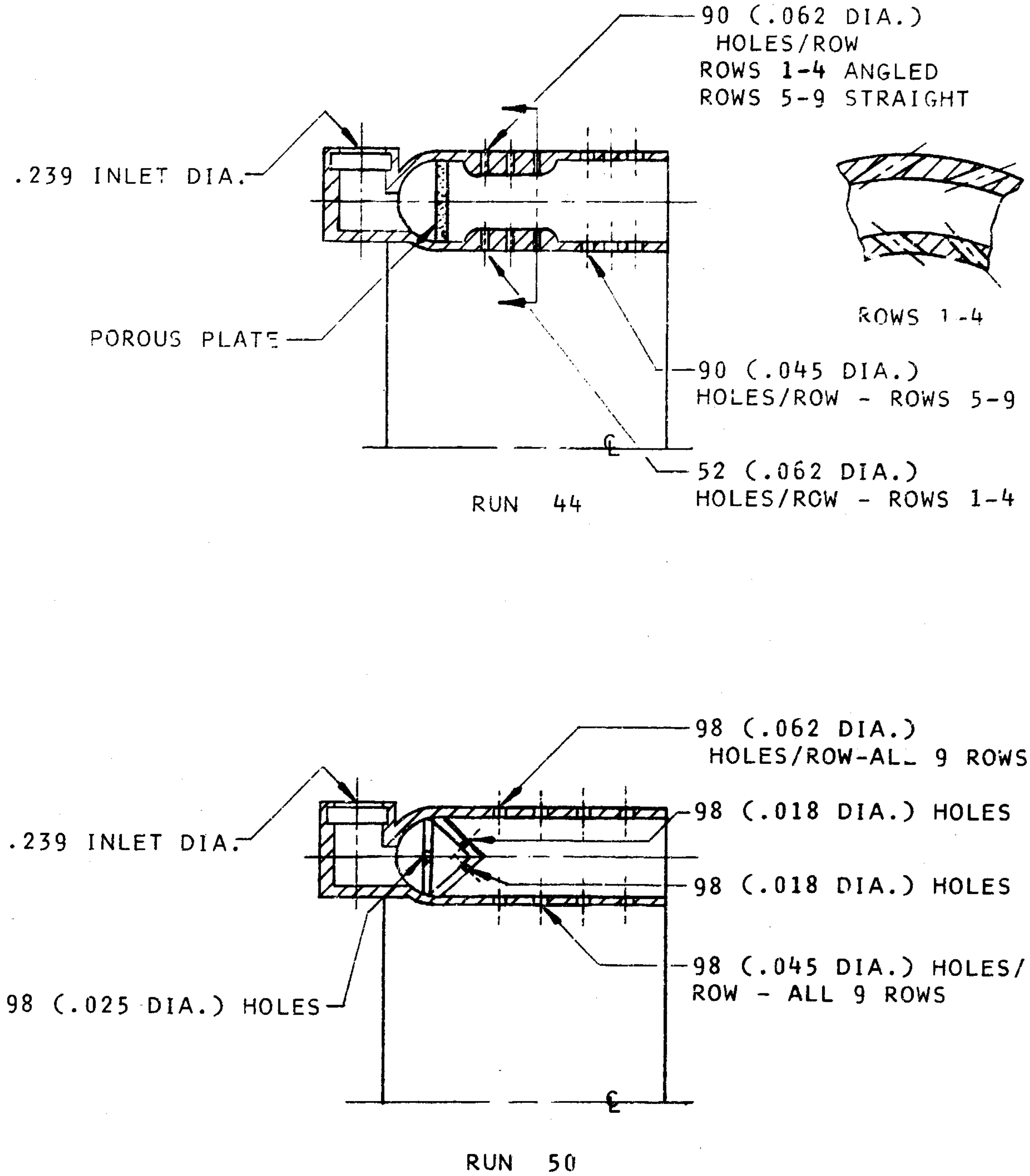
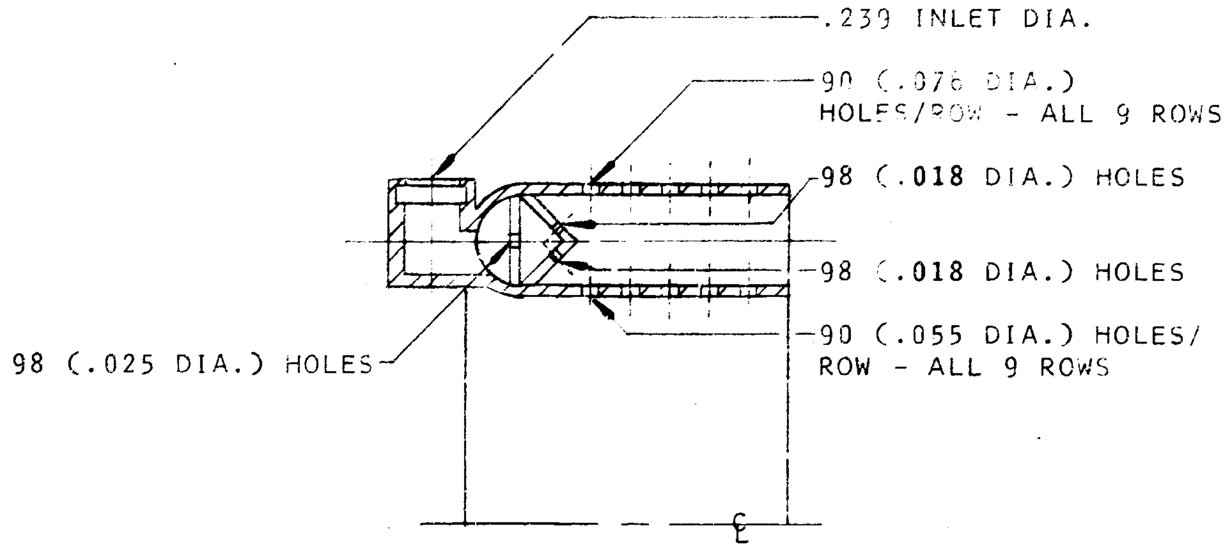
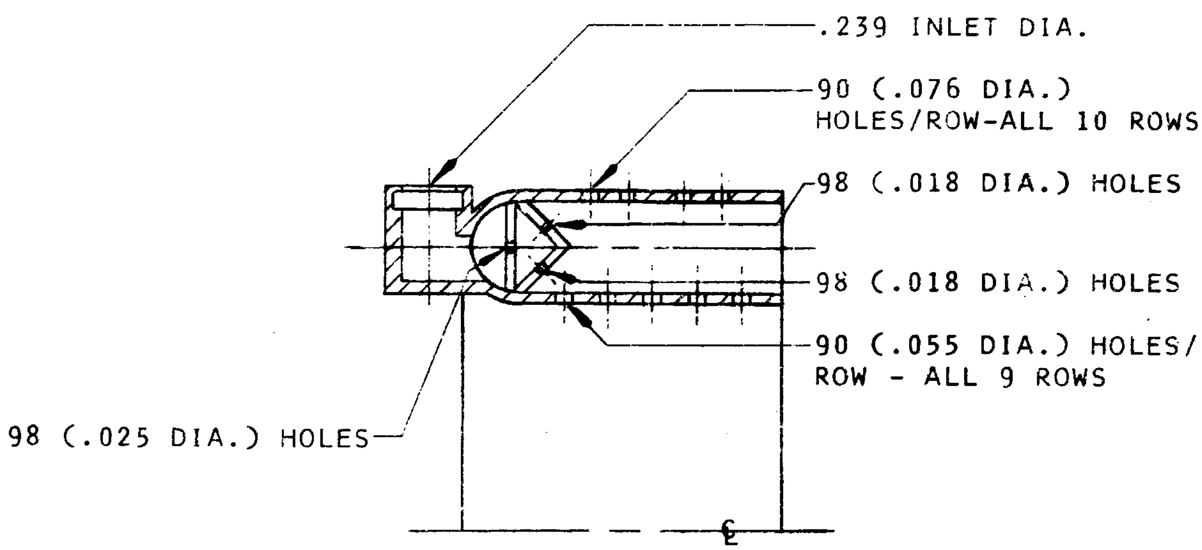

90 (.062 DIA.)
HOLES/ROW
ROWS 1-4 ANGLED
ROWS 5-9 STRAIGHT
.239 INLET DIA.
ROWS 1-4
POROUS PLATE
90 (.045 DIA.)
HOLES/ROW - ROWS 5-9
52 (.062 DIA.)
HOLES/ROW - ROWS 1-4
RUN 44
98 (.062 DIA.)
HOLES/ROW-ALL 9 ROWS
98 (.018 DIA.) HOLES
.239 INLET DIA.
98 (.018 DIA.) HOLES
98 (.045 DIA.) HOLES/
ROW - ALL 9 ROWS
98 (.025 DIA.) HOLES
RUN 50

FIG. 6

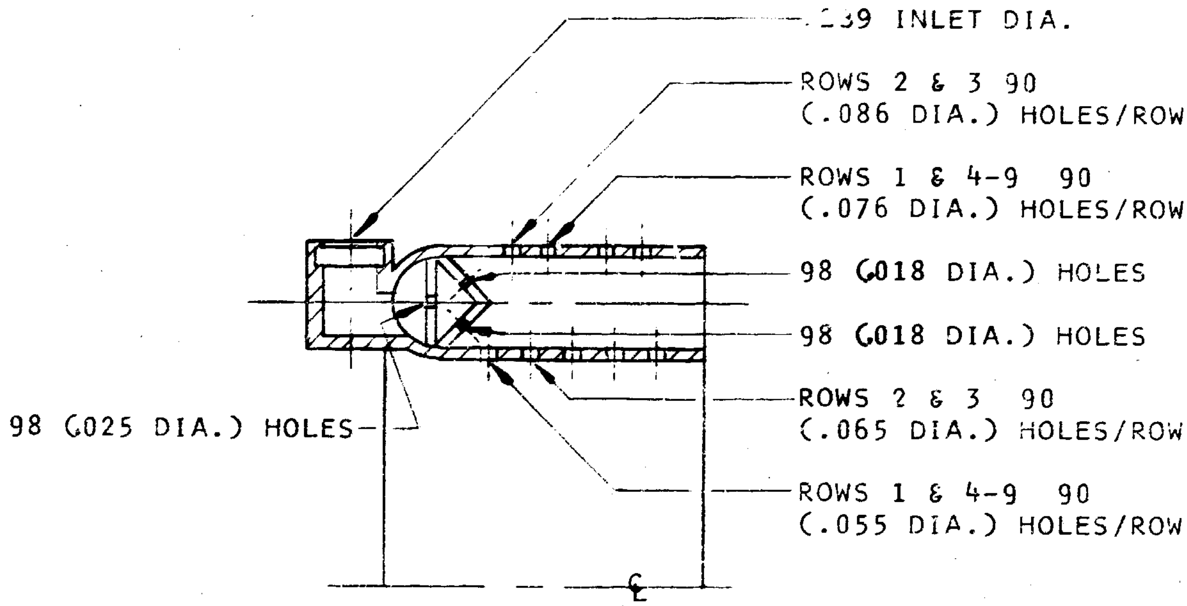


RUN 54



RUN 55

FIG. 7



RUNS 56 THROUGH 65

FIG. 8

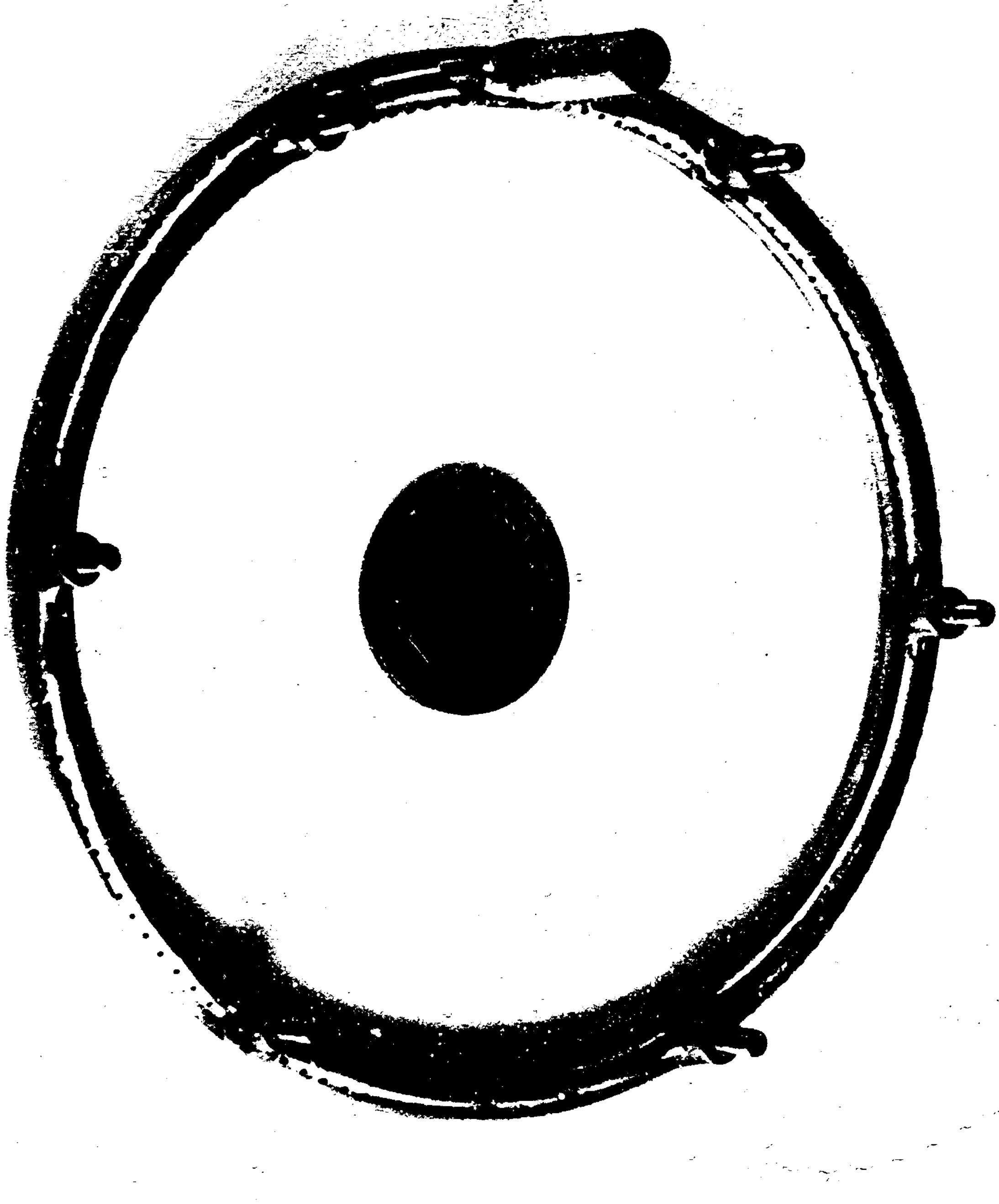


FIGURE 9 - FUEL NOZZLE

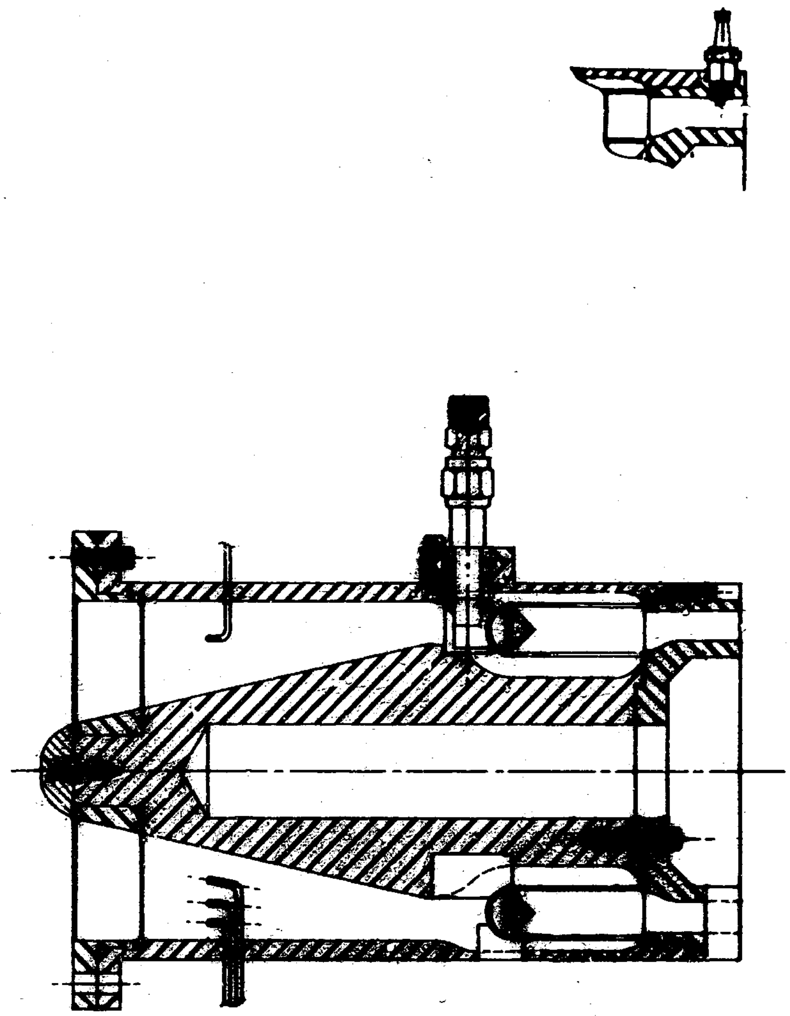


FIGURE 10 - BURNER ASSEMBLY

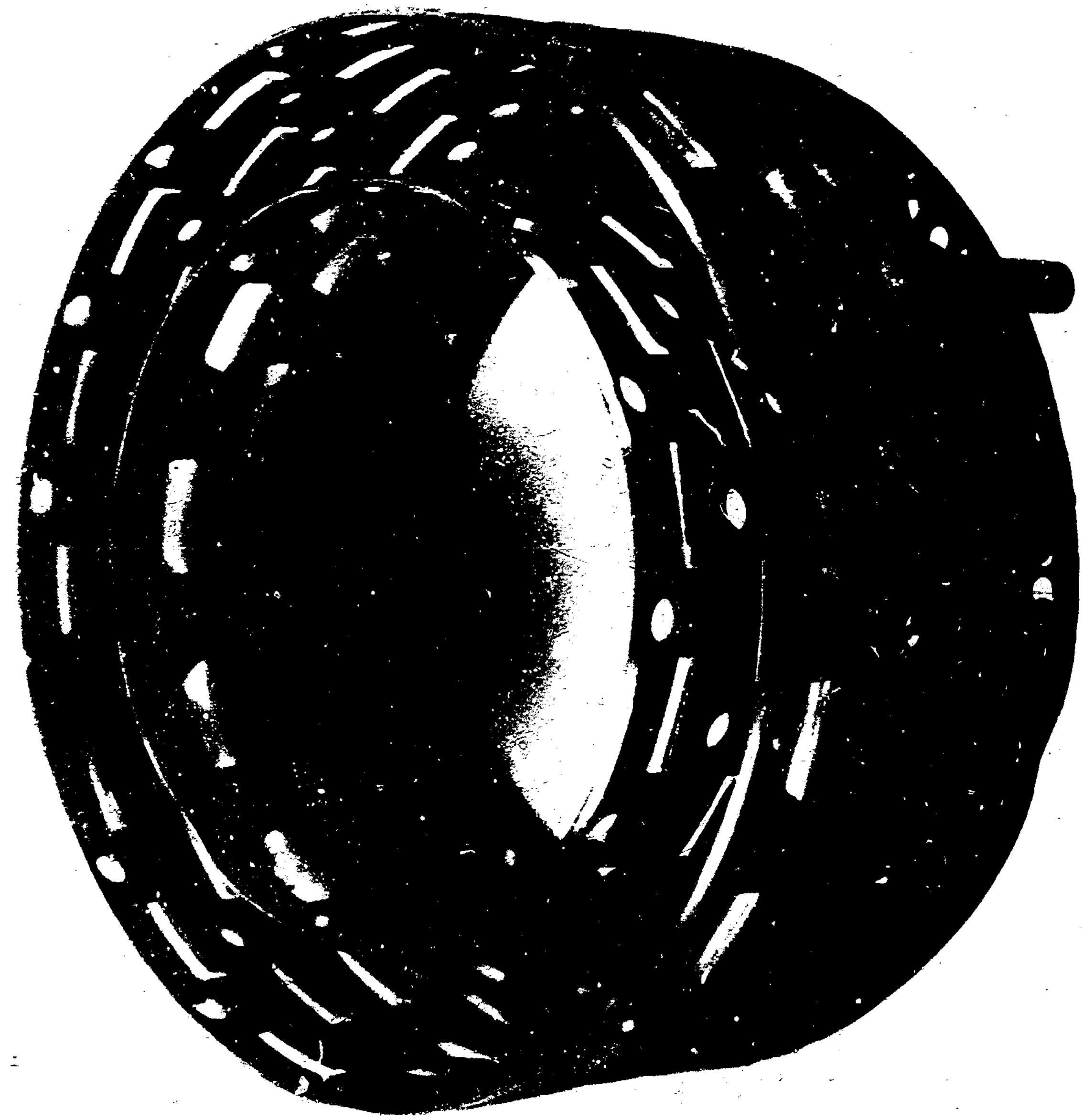


FIGURE 11 - LOUVERED LINER

CIRCUMFERENTIAL TEMPERATURE

DISTRIBUTION

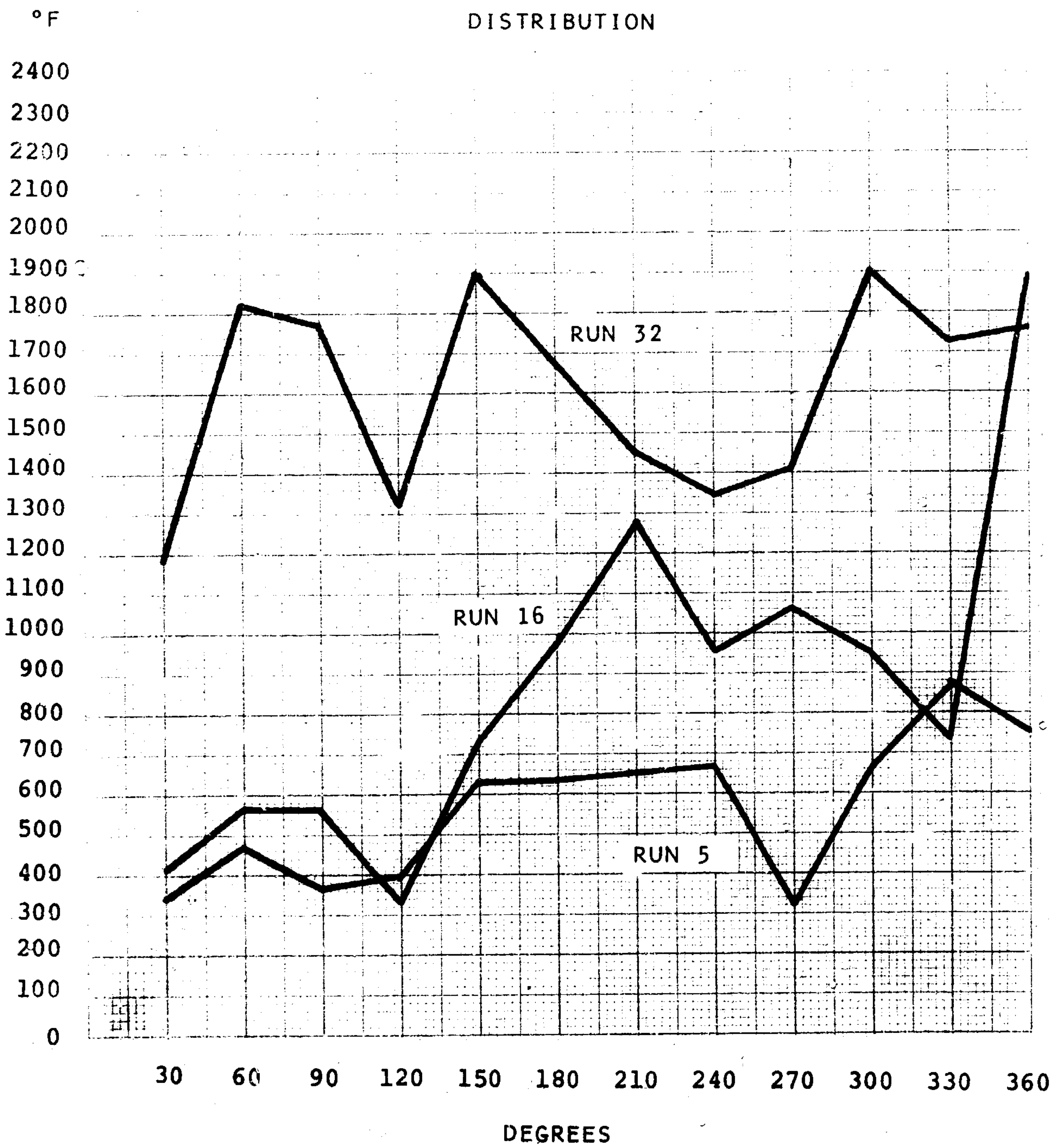


FIG. 12

CIRCUMFERENTIAL TEMPERATURE

DISTRIBUTION

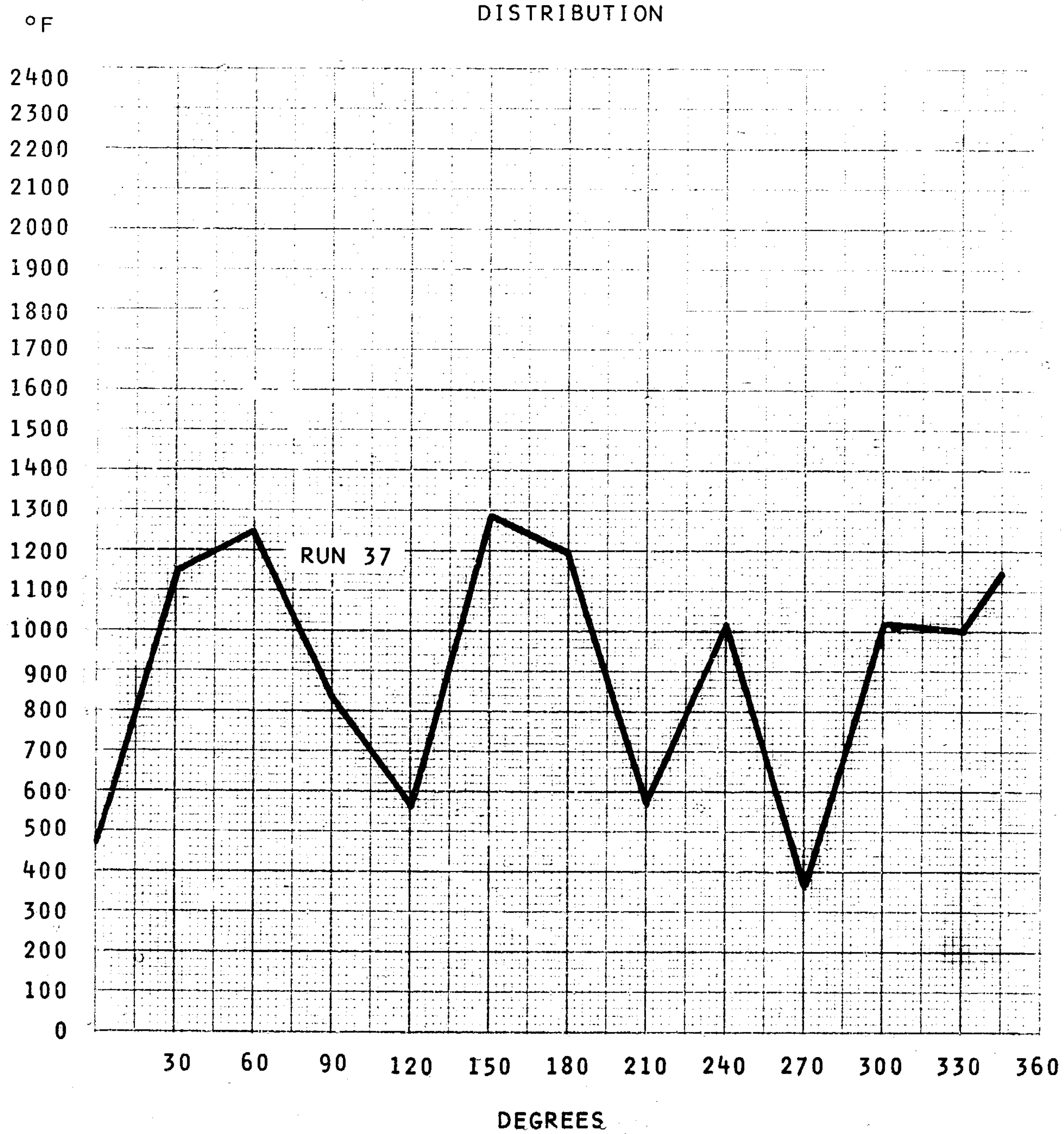


FIG. 13

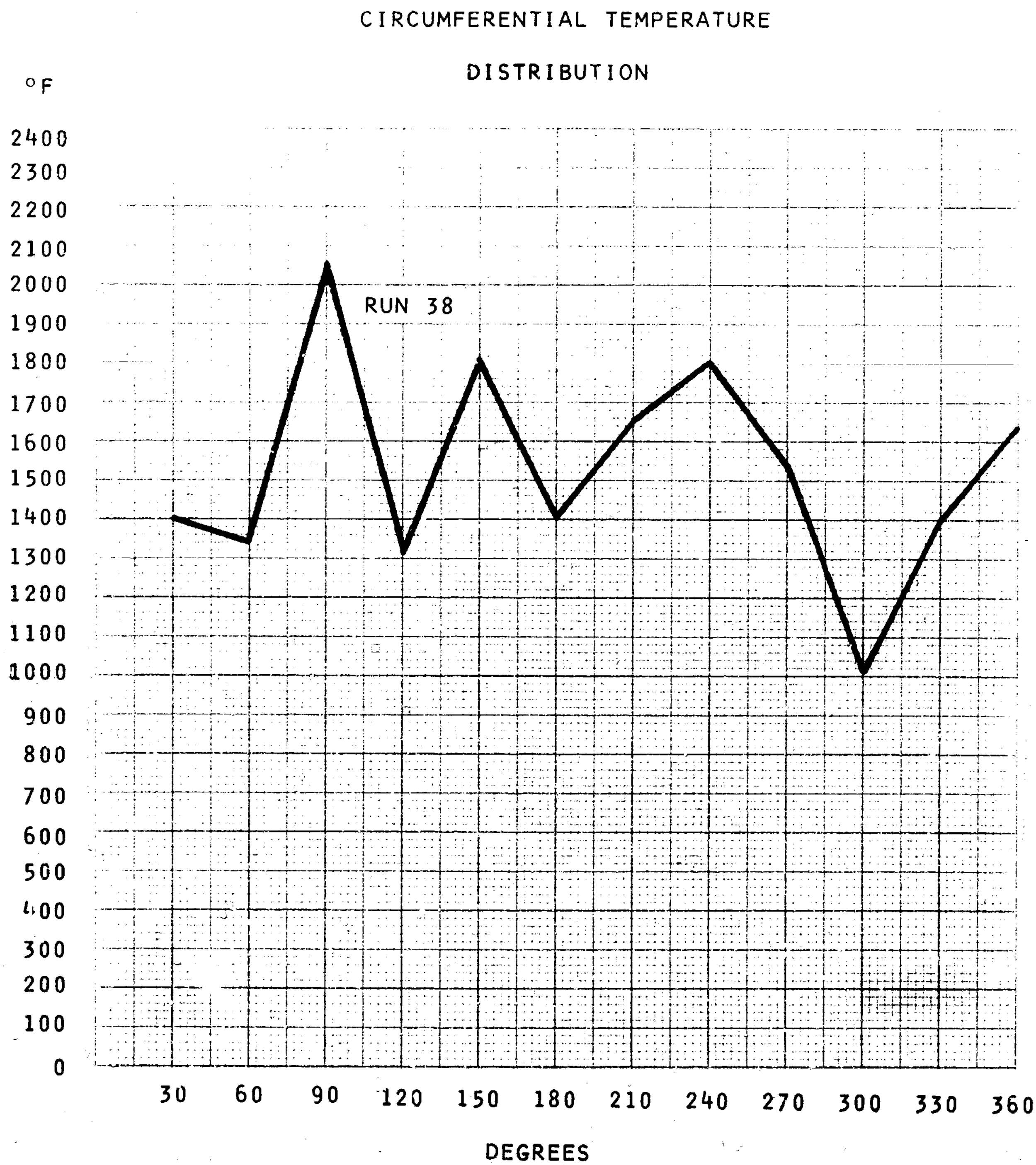

CIRCUMFERENTIAL TEMPERATURE
DISTRIBUTION
°F
2400
2300
2200
2100
2000
1900
1800
1700
1600
1500
1400
1300
1200
1100
1000
900
800
700
600
500
400
300
200
100
0
RUN 38
30
60
90
120
150
180
210
240
270
300
330
360
DEGREES

FIG. 14

CIRCUMFERENTIAL TEMPERATURE

DISTRIBUTION

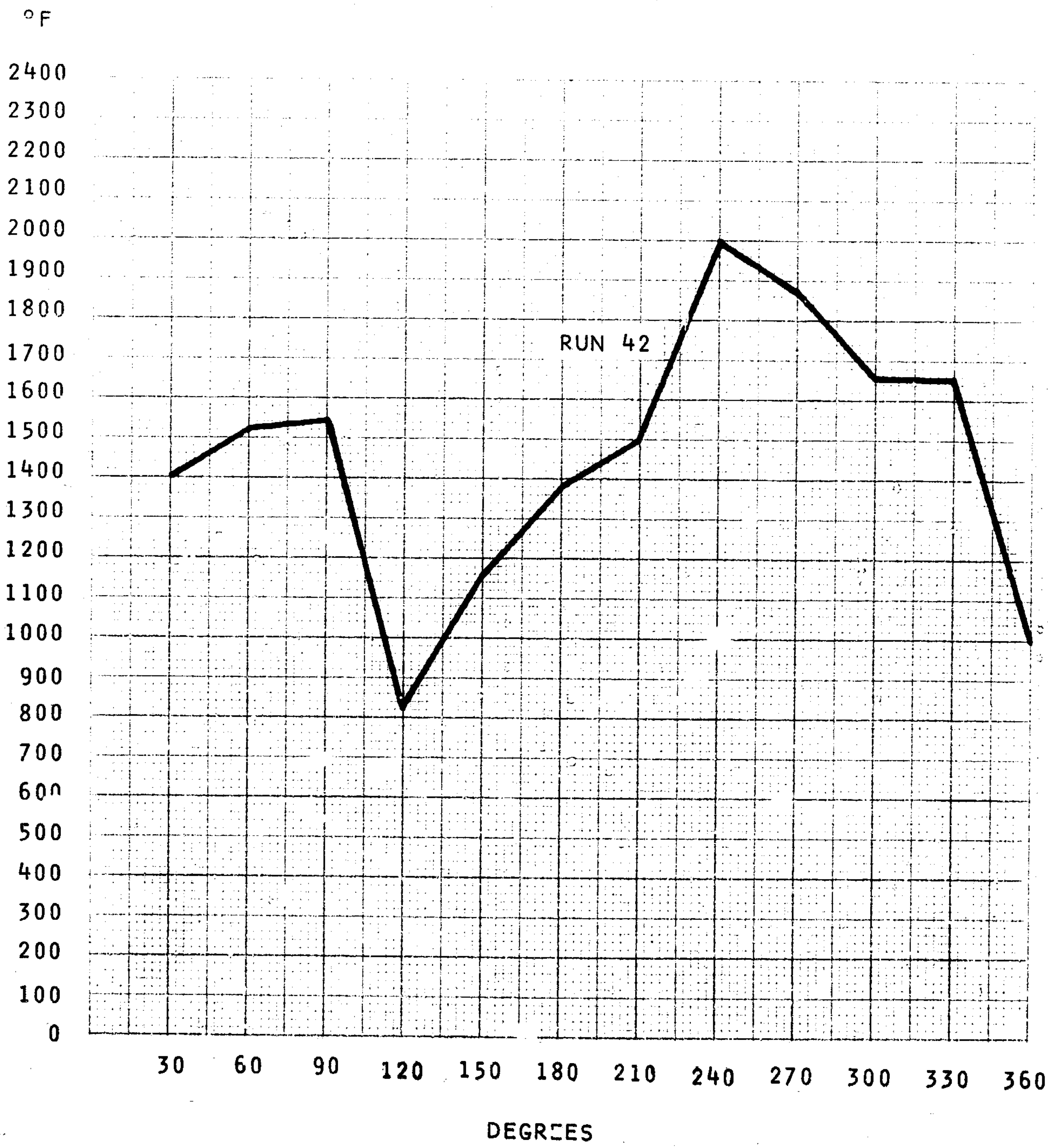


FIG. 15

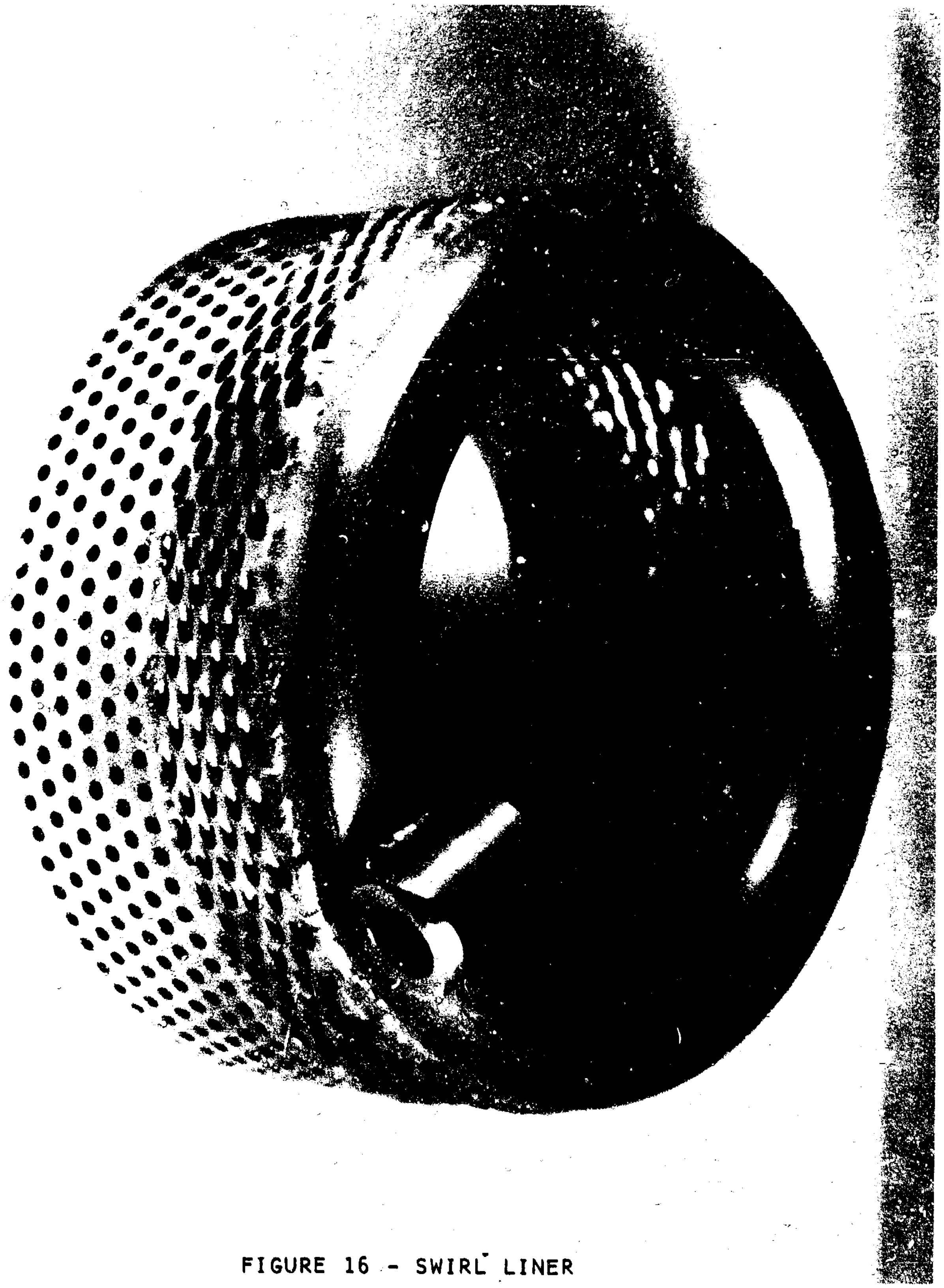


FIGURE 16 - SWIRL LINER

CIRCUMFERENTIAL TEMPERATURE

DISTRIBUTION

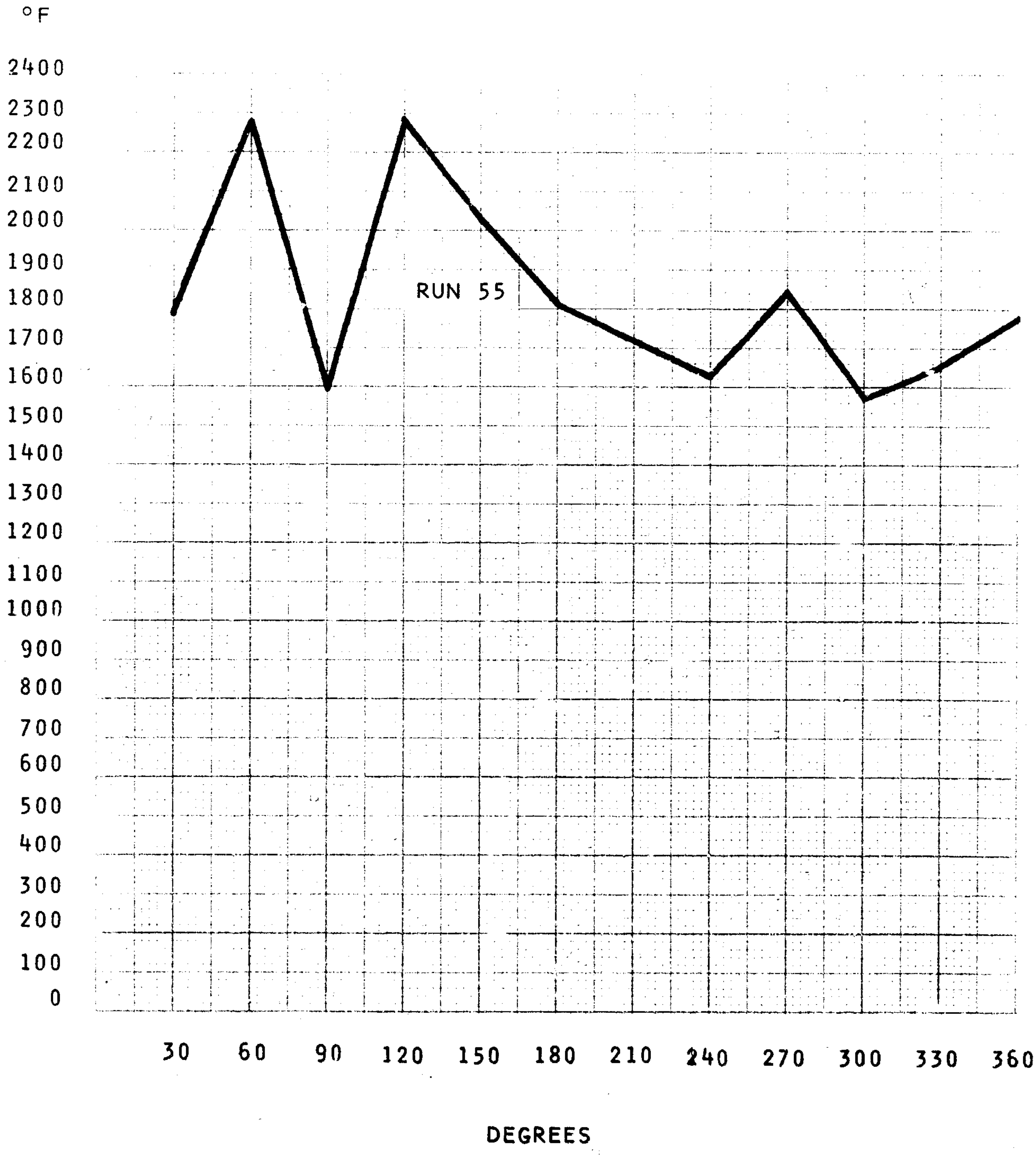


FIG. 17

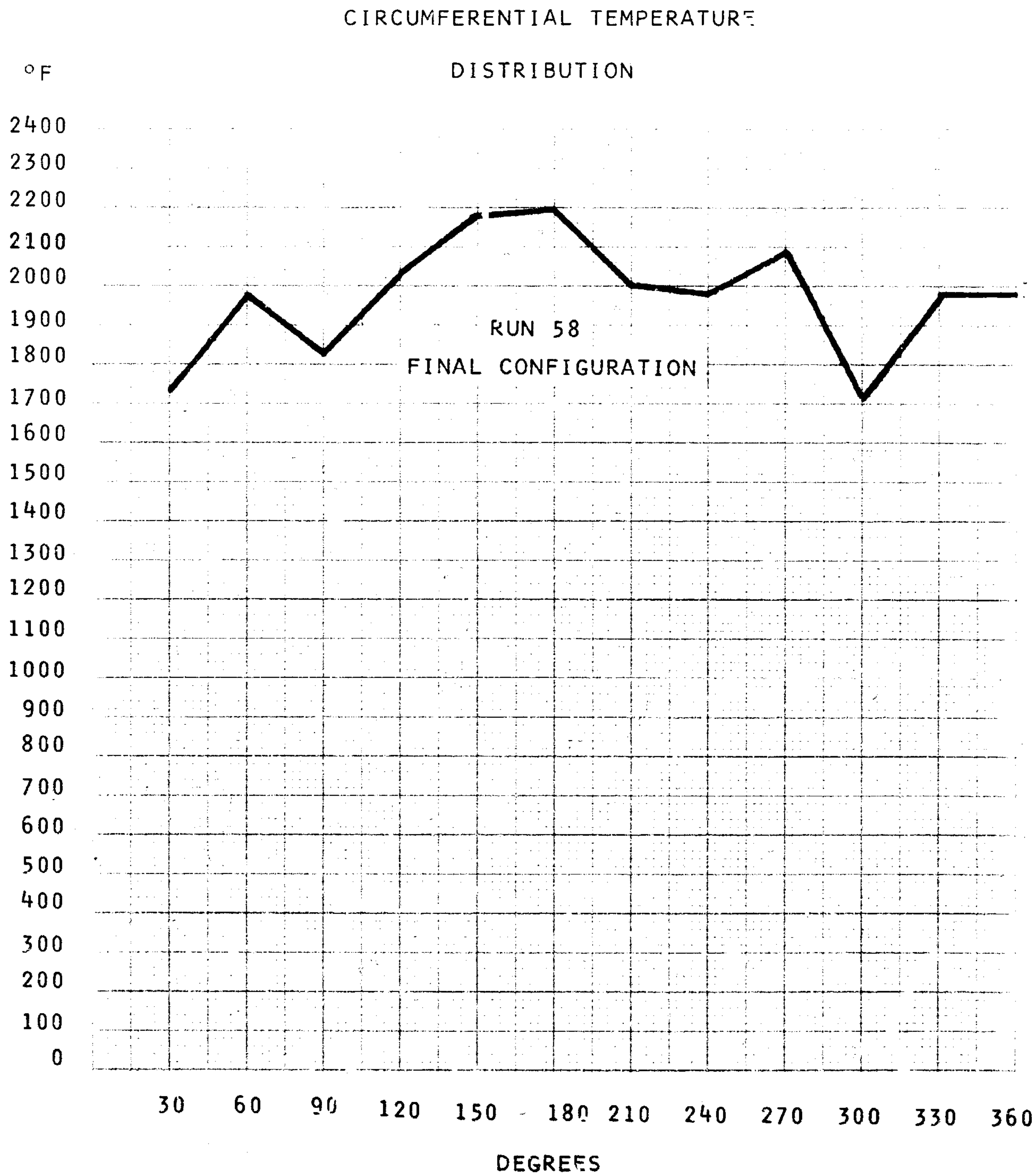

CIRCUMFERENTIAL TEMPERATURE
DISTRIBUTION
°F
2400
2300
2200
2100
2000
1900
1800
1700
1600
1500
1400
1300
1200
1100
1000
900
800
700
600
500
400
300
200
100
0
RUN 58
FINAL CONFIGURATION
30
60
90
120
150
180
210
240
270
300
330
360
DEGREES

FIG. 18

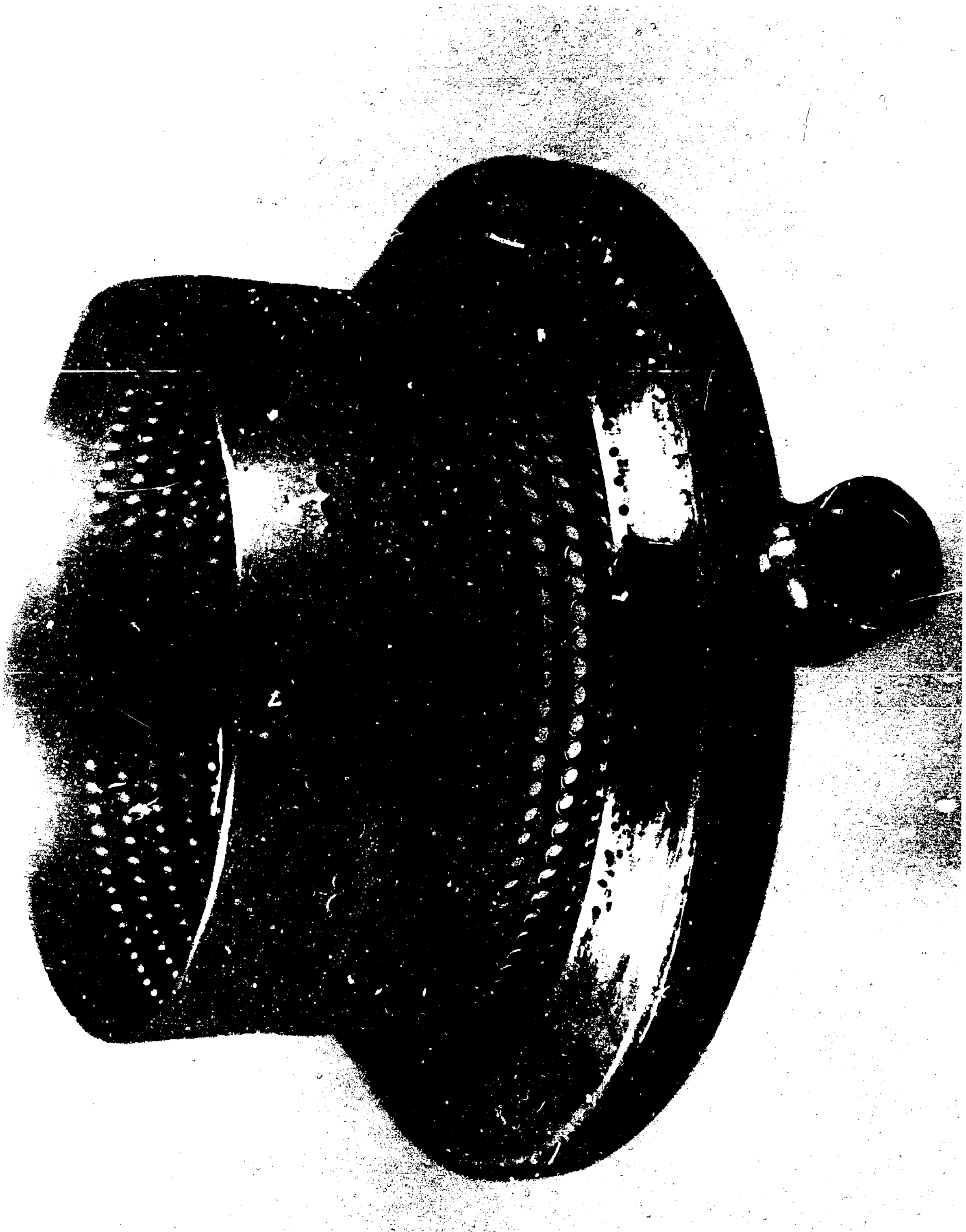


FIGURE 19 - FINAL COMBUSTOR LINER WITH OUTER LINER REMOVED

ENDURANCE RUN - RUN 60-62

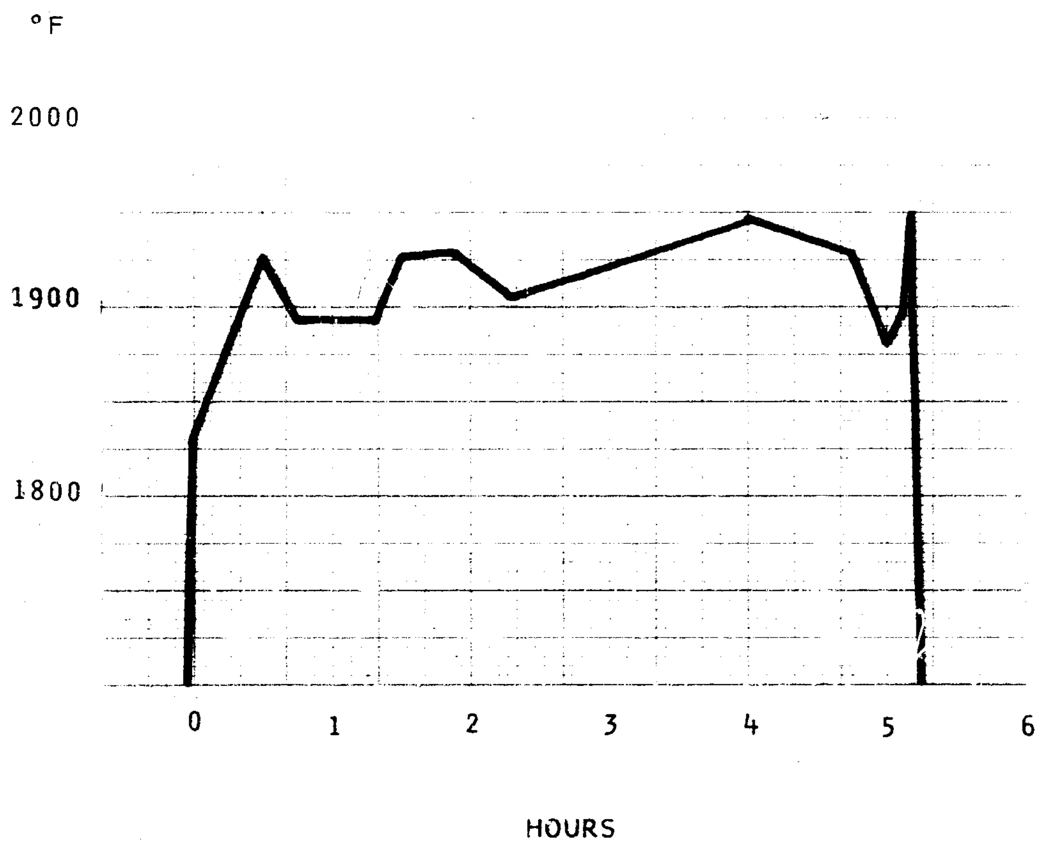


FIGURE 20

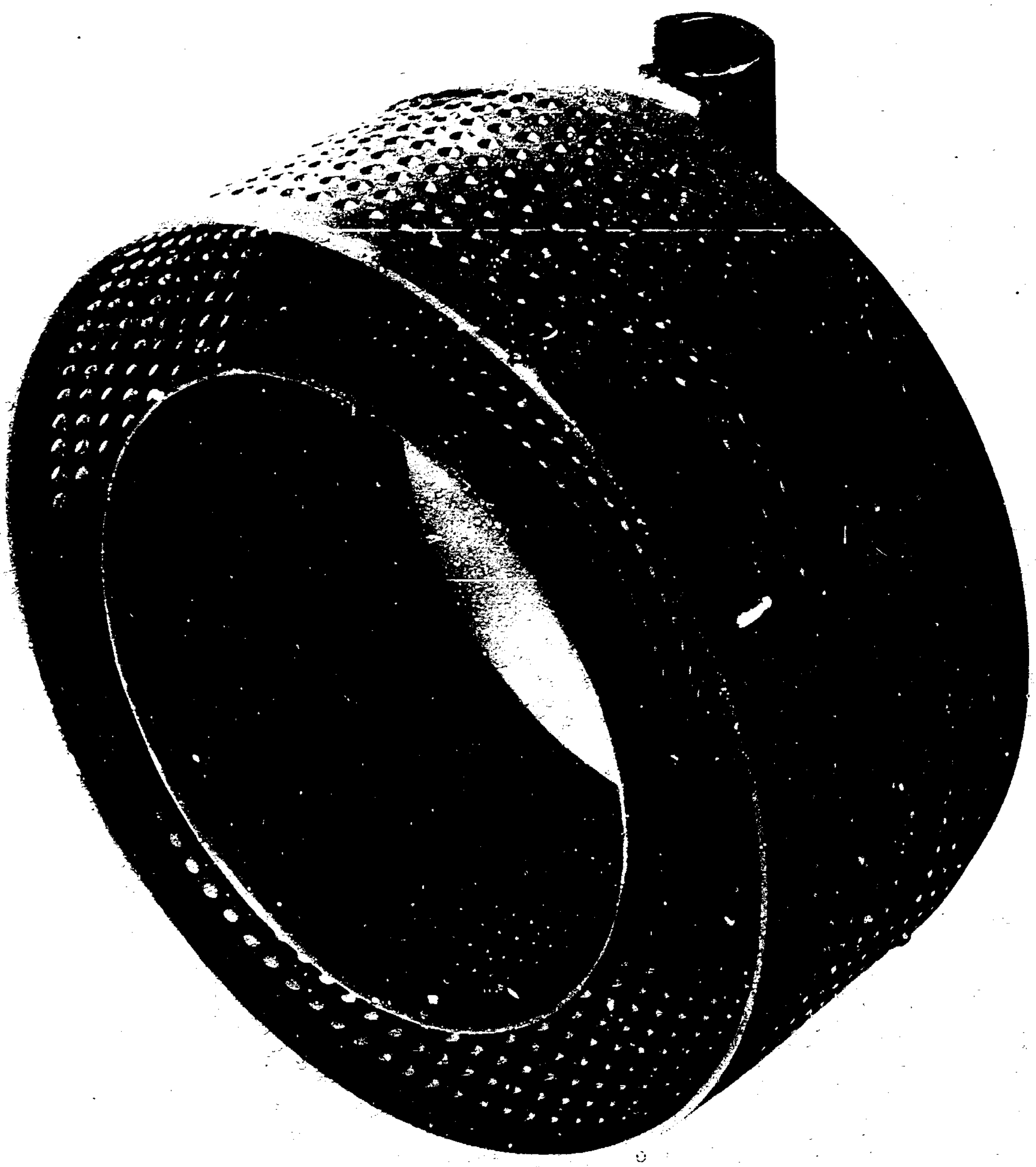


FIGURE 21 - FINAL COMBUSTOR LINER
BEFORE ENDURANCE RUN

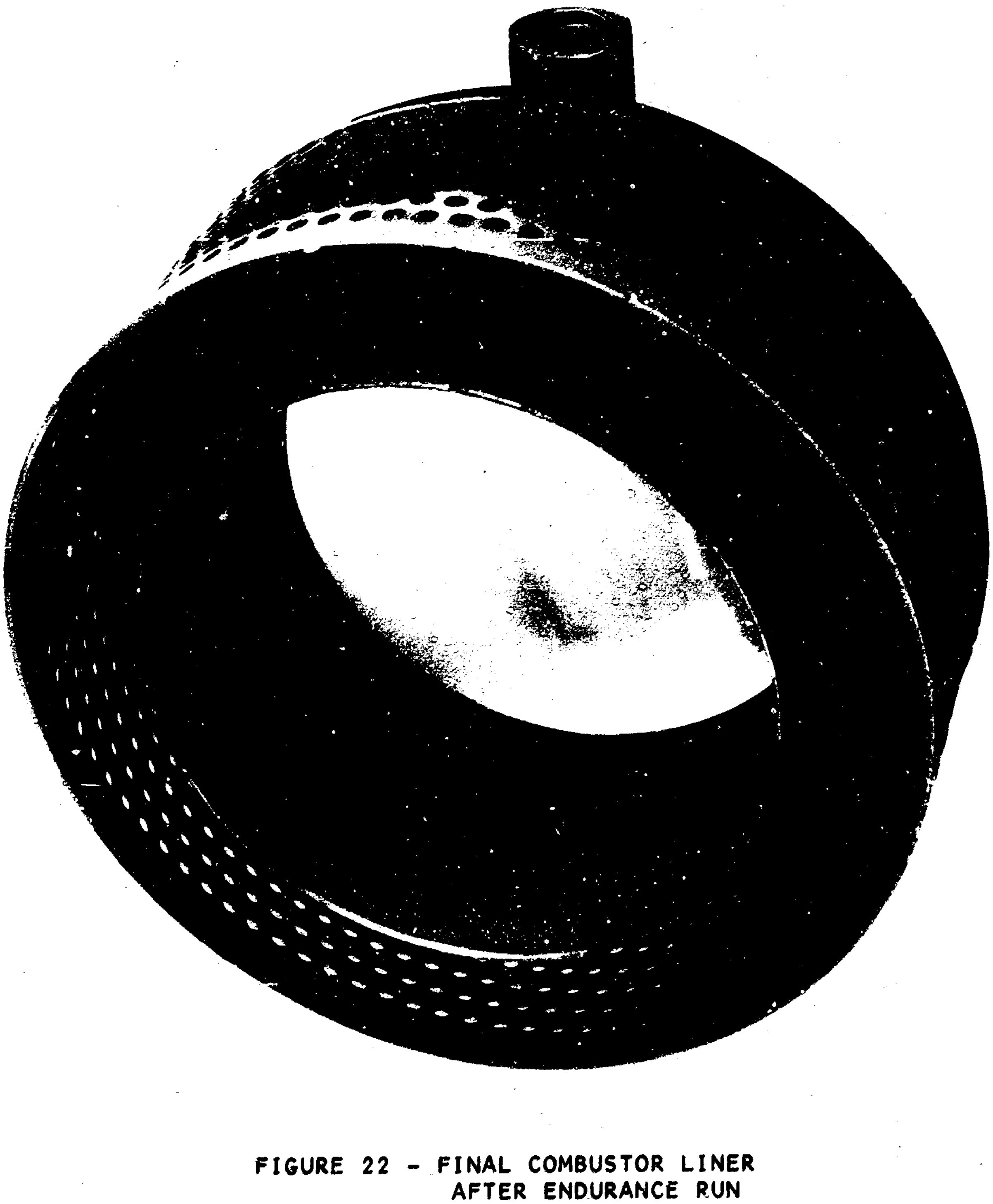


FIGURE 22 - FINAL COMBUSTOR LINER
AFTER ENDURANCE RUN

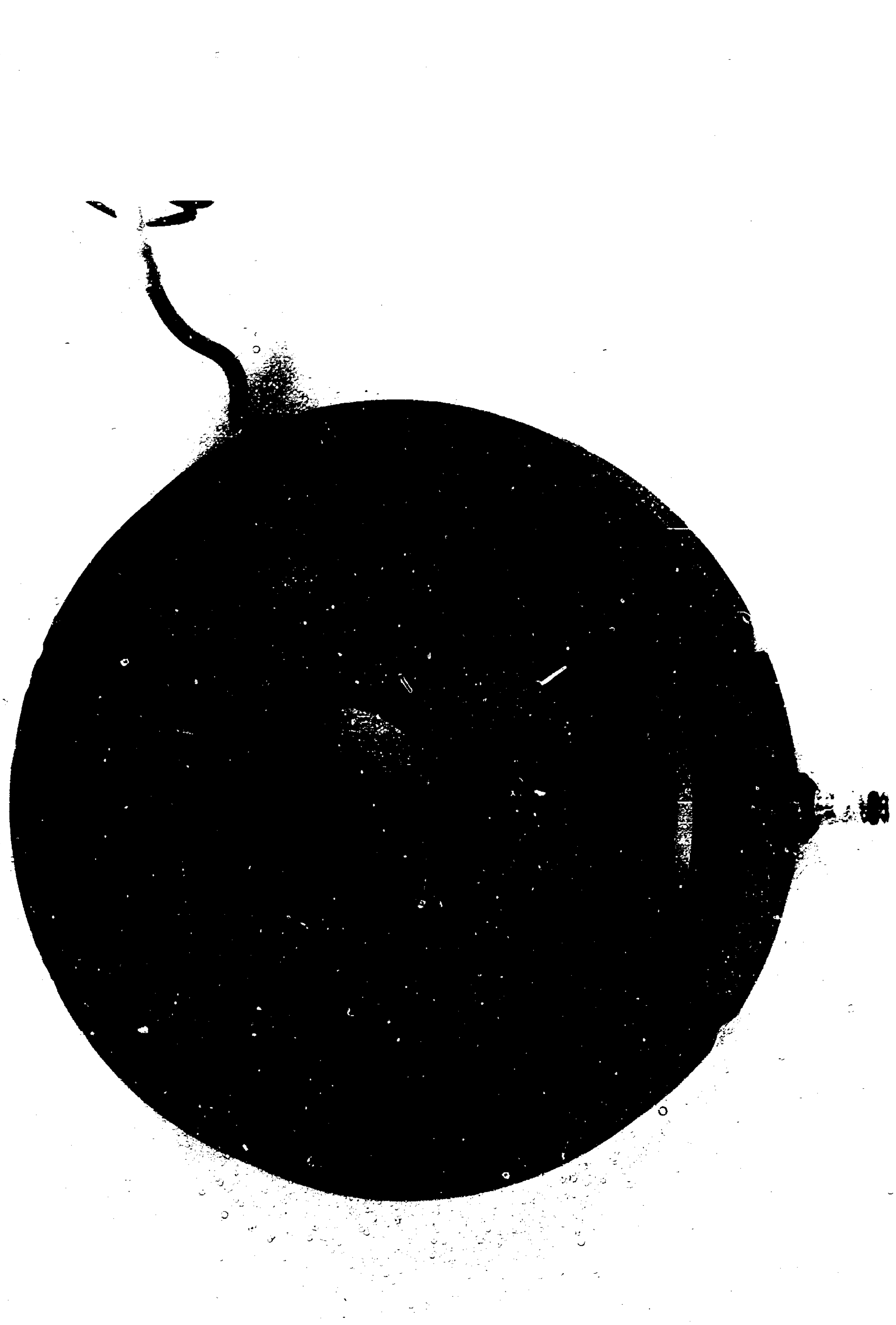


FIGURE 23 - SIMULATED TURBINE NOZZLE

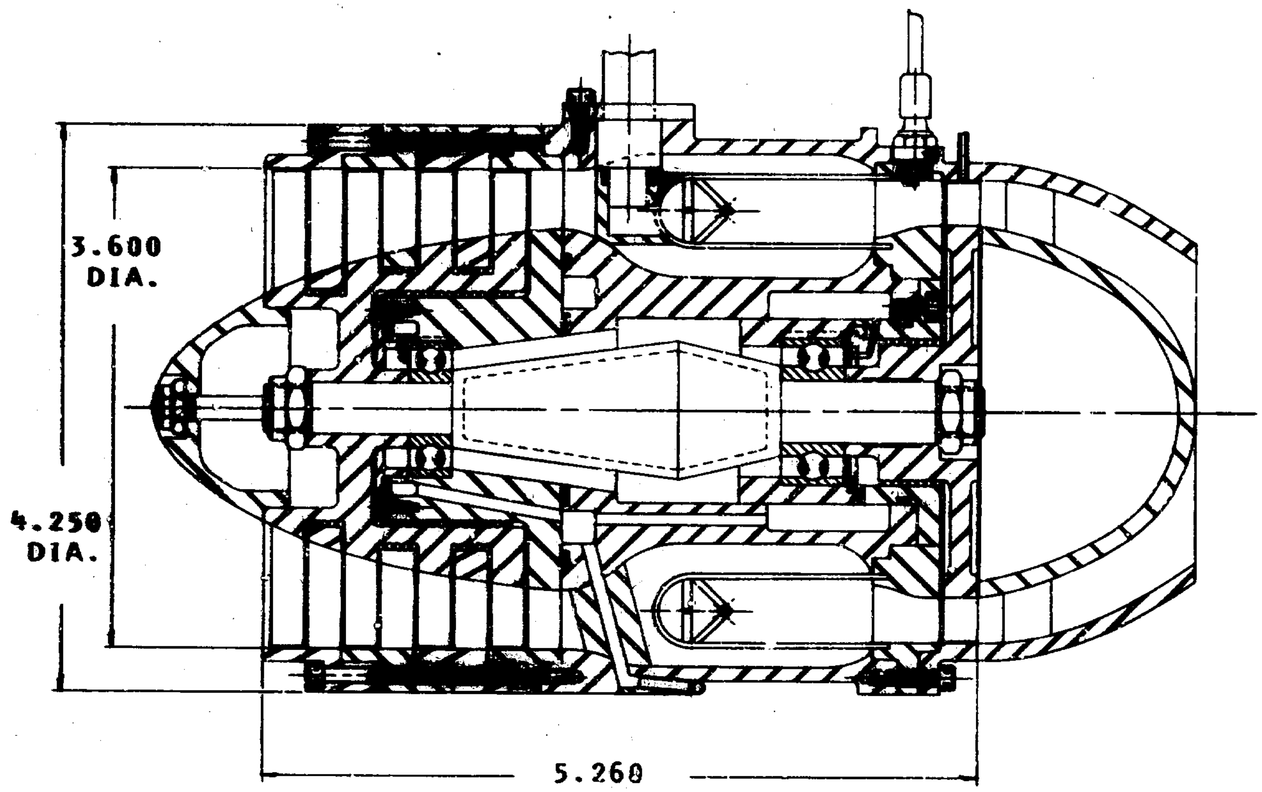


FIG. 24 - PROPOSED ENGINE ASSEMBLY